REPORT No. 297

THE REDUCTION OF OBSERVED AIRPLANE PERFORMANCE TO STANDARD CONDITIONS

By WALTER S. DIEHL
Bureau of Aeronautics

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SUMMARY

This report shows how the actual performance of an airplane varies with air temperature when the pressure is held constant. This leads to comparatively simple methods of reducing observed data to standard conditions. The new methods which may be considered exact for all practical purposes, have been used by the Navy Department for about a year, with very satisfactory results.

The report also contains a brief historical review of the important papers which have been published on the subject of performance reduction, and traces the development of the standard atmosphere.

INTRODUCTION

The analysis of observed airplane performance is greatly complicated by the fact that the power available and the power required in flight do not vary according to the same law. At constant R. P. M. the engine power is a function of air pressure and air temperature independently, so that the power developed at a given density depends on the combining pressure and temperature. On the other hand, the power absorbed by the propeller at constant R. P. M., and the power required for horizontal flight are constant at a given density regardless of the combining pressure and temperature. Since, on two successive days the pressures and temperatures at given true altitudes or at given densities may vary widely from the average or standard values, it follows that observed performance and in particular the rates of climb at given true altitudes or at given densities also vary widely with the atmospheric conditions. Climbs made on different days rarely agree. The problem is to find some method of reducing observed data to a standard condition by applying corrections such that, regardless of the atmospheric conditons under which the flight tests are made, the reduced data are always in agreement.

There are two basic conditions which must be specified before any accurate method of reduction can be formulated. The first and most important condition is a Standard Atmosphere or a definite set of relations between pressure temperature, density, and altitude to which the reduced performance may be referred. Arbitrary but definite relations must be used, since the performance is determined by the pressure and temperature and not by the geometric or true altitude. The second condition is the variation of engine power with pressure and temperature.

A standard atmosphere having been adopted, there are two methods of attack. One may assume various relations between engine power and atmospheric pressure and temperature and by trial and error find the relation which gives the best agreement in reducing observed performance, or one may find from test stand data the effect of pressure and temperature on engine power and from this calculate the variation in performance due to variation in atmospheric conditions. The latter method will be used in the present study.

Before taking up the development of an accurate method of performance reduction, a brief historical review of the subject will be made. This review will quote extracts from the more important papers in order to illustrate the steps that have been taken in the improvement of performance reduction, in the adoption of the standard atmosphere, and in the matter of variation of power with altitude.

The terms "pressure altitude" and "density altitude" will occur frequently. These are simply the altitudes in the standard atmosphere defined by a given pressure, or a given density, respectively.

HISTORICAL REVIEW

Prior to 1915 the performance testing of airplanes was confined to the measurement of high speeds at ground level and to the observation of partial climbs, usually between certain specified aneroid heights such as 3,000 to 5,000 feet, and these climbs were often made with service altimeters. About 1915 the Royal Aircraft Factory made the first definite attempt to place airplane performance testing on a scientific basis. Little is known concerning the initial efforts in this line, but the first paper of any importance on the subject is by Captain H. T. Tizard, R. F. C. This paper, entitled "Methods of Measuring Aircraft Performance," was read before the Royal Aeronautical Society March 7, 1917, and published in the April–May–June issue of the Royal Aeronautical Journal. It opens with a brief historical review followed by a very clear statement of the necessity for reducing observed performance to some standard condition before comparisons are made. The standard condition then employed by the R. A. F. and R. F. C. was obtained from a table of mean atmospheric pressures, temperatures, and densities at various heights, based on observations by Mr. W. H. Dines. A standard density corresponding to 1.221 kg/m^3 had been adopted and a table of relative densities against altitude prepared from Dines's data. It is interesting to note that the resultant sea-level density ratio 1.026 thus obtained was used more or less generally five years later, although the subject came up in the discussion of Captain Tizard's paper. In view of the general use of the present standard atmosphere the following quotation from the discussion is of historical interest: "Lieut. G. H. Millar, R. N. V. R., said that in his opinion it was a pity that the standard atmosphere which had been adopted was a purely empirical one; he would have preferred one based on a given temperature and pressure at sea level, and a uniform rate of fall of temperature * * *. The advantage of such a standard over the empirical one was that it could be calculated at any time by remembering two constants. He also thought that the unit of density should certainly be the density at zero height for the standard atmosphere adopted." Lieutenant Millar's remarks apparently anticipate by several years the linear temperature gradient usually credited to Toussaint and now used in the standard atmosphere.

In this connection a linear temperature gradient was proposed in the appendix to a report by H. Glauert and S. B. Gates, entitled "Prediction of the Performance and Longitudinal Stability of an Aeroplane, Including the Estimation of the Effect of Small Changes in the Design." This report, dated March, 1917, was published as R. & M. No. 324 by the British Advisory Committee for Aeronautics. According to the author's notes, a linear temperature gradient was also used by Prof. E. P. Warner in a lecture on Airship Design at Massachusetts Institute of Technology, May 28, 1918. The linear temperature gradient now in use $\frac{dT}{dh}=0.0065$ °C/m is based on observations by Professor Gamba at the Padua Observatory between 1906 and 1916. Professor Gamba's data are given in a report published by the experimental section of the Italian Air Service in September, 1918. This report and the linear gradient are discussed in an article entitled "Lois Experimentales des Variations de la Pression Barometrique et du Poids Specifique de l'air avec Altitude," by R. Soreau in the November 1–15 issue of l'Aerophile. Soreau proposes, however, to use a rather complicated empirical formula, which appears to have no advantage other than that of reasonable agreement with observed data. Toussaint's proposal to use a linear temperature gradient was first issued by the S. T. Aé. in March, 1920, as the "Draft of Interallied Agreement on Law Adopted for the Decrease of Temperature with Increase of Altitude." This draft is rather brief, but all essential information is given. It was followed by a more complete definition, together with a quantity of supporting data, in the Bulletin Trimestriel des Etudes, issued by S. T. Aé. in April, 1920. Toussaint's proposal was officially adopted by the S. T. Aé., beginning on May 1, 1920. During the same year England, Belgium, and Spain agreed to adopt the proposal, and in 1921 it was adopted with one very slight modification by the National Advisory Committee for Aeronautics for use in the United States. Before the adoption in this country a very thorough investigation by the Weather Bureau showed that it was more desirable to use an isothermal temperature of −55° C. instead of the −56.5 proposed by Toussaint. The difference is negligible in so far as it ever effects any performance reduction. The data supporting this point are given and discussed by Willis Ray Gregg in N. A. C. A. Technical Report No. 147, "Standard Atmosphere" (1922).

Returning to Captain Tizard's paper, the proposed method of reduction was on a density basis. Aneroid altitudes were plotted against time and rates of climb in "aneroid feet" were read from the slope of the curve. These rates of climb were then corrected to "true" rates by multiplying by a correction factor corresponding to the difference between the observed temperature and the +10° C. constant temperature used in calibrating all aneroids at that time. These "true" rates of climb were then plotted against the standard density heights corresponding to each pressure (aneroid) and temperature reading. Speeds were also plotted against density altitudes after correcting the observed readings for instrumental error and converting to true speeds.

The second paper on performance reduction appears to be Captain Toussaint's "Note sur les Methodes et les Calculs d'essais des Avions Nouveaux," a series of lectures given at l'Ecole Superieure d'Aeronautique at Paris, January and April, 1918. These lectures were published in mimeographed form by the S. T. Aé. and translated by the Technical Publications Department at McCook Field. A limited number of blue-print copies of this translation, dated September, 1918, were distributed as confidential documents. The method of reduction described by Toussaint is on a density basis quite similar to that outlined by Tizard, except that the standard atmosphere then used was based on Radau's law connecting pressure temperature and pressure

$$\frac{\Delta T}{\Delta P}=\left(\frac{T_o-T}{P_o-P}\right)=0.08 \tag{1}$$

where T is in °C and P in mm Hg. It is of interest to note that Radau's law dates from 1864. Toussaint's method was used extensively in the United States prior to about 1923.

The first published method of performance reduction on the pressure basis appears to be that outlined in a paper by D. H. Pinsent and H. A. Renwick, "The Variation of Engine Power with Height," which was published in March, 1918, as R. & M. No. 462 by the British Advisory Committee for Aeronautics. The following extract from the summary of this report is of great interest: "The experiments indicate in all cases that the B. HP. can better be expressed as a function of the barometric pressure only, instead of the previous and customary method where it was assumed to be a function of density * * * The above conclusion makes it necessary to reconsider the methods at present in use of reducing climbs and speeds at heights to a standard basis for comparison. It has been customary hitherto in comparing the performances of two airplanes as regards speed to compare their true air speeds at the same density; and as regards climb, to compare their times to climb from one given density to another. If engine power can be better expressed as a function of pressure only, this method of comparison is unduly favorable to airplanes tested in warm weather, and vice versa."

Bairstow in his Applied Aerodynamics, Longmans, Green & Co., 1920, gives a method of reduction on a density basis that is practically the same as Tizard's, but his remarks on the method include the following statement (p. 430); "The standard method of reduction of British performance trials has up to the present date been based on the assumption that the engine horsepower depends only on the density. Questions are now being raised as to the strict validity of this assumption, and the law of dependence of power on pressure and temperature is being examined by means of specially conducted experiments."

The experiments to which Bairstow refers were made during 1919 and 1920, but they were not reported until 1924. The results are given in a paper entitled "Variation of Engine Power with Height," by H. L. Stevens, and published by the British Aeronautical Research Committee in August, 1924, as R. & M. No. 960. This report contains a number of climbs reduced on both the density basis and the pressure basis, and the conclusion given in the summary is: "For performance reduction it is better to assume that the engine power is proportional to some power of the pressure ($p^{1.05}$ in this case) than some power of the density." This conclusion is also incorporated in the text and with more emphatic wording: "The direct measurements of performance indicate that for the reduction of climbs to standard conditions a pressure law is definitely better than a density law." An appendix giving brief but complete outlines of the two methods of performance reduction contains under the pressure method the very interesting statement: "* * * It follows that the top speed of level flight of an air plane is such that the indicated speed at a definite aneroid height is independent of the temperature."

A second report, "The Variation of Engine Power with Height," by H. M. Garner and W. G. Jennings, was issued by the British Aeronautical Research Committee in September 1924, as R. & M. No. 961. The conclusion based on flight tests with a torque meter was: "The experiments show that the engine power is very nearly a function of the pressure only, except for low heights, where it depends to a certain extent on the temperature."

In a paper, "Engine Performance and the Determination of Absolute Ceiling," which was published as N. A. C. A. Technical Report No. 171 in September, 1923, the author showed that B. HP. varied as $p^{1.15}$ when temperature and R. P. M. are held constant and as $T^{-0.50}$ when pressure and R. P. M. are held constant. At this time the density method of performance reduction was used generally in the United States, as elsewhere, and while it was realized that the pressure method apparently gave more consistent results than the density method, there was considerable opposition to a change. In view of the desirability of having a definite and easily followed method available, the author prepared a simplified approximate density method, and Prof. E. P. Lesley prepared a more elaborate density method of reduction. These methods were published together in 1925 as N. A. C. A. Technical Report No. 216, "The Reduction of Airplane Flight Test Data to Standard Atmosphere Conditions."

A paper by R. S. Capon, "Note on the Reduction of Performance Tests to the Standard Atmosphere," was published by the British Aeronautical Research Committee in January, 1927, as R. & M. No. 1080. Analysis was made of 12 pairs of climbs using special test instruments corrected for temperature effects. The conclusion regarding variation of power with height as given in the summary is: "The law specified by the climbs with corrected instrument calibrations is $f\ (p^{2/3}\rho^{1/3})$. It is indicated that the mean error of the rates of climb determinations is considerably reduced by the use of corrected instruments and is such that the adoption of a power factor which is a function of pressure only will not do justice to the accuracy of the test data." An appendix contains a brief outline of the proposed method of reduction, which states that if any two of the equations:

$$\frac{V'}{V}=\frac{v'_c}{v_c}=\frac{RPM'}{RPM}=\frac{f'HP}{fHP}=\sqrt{\frac{\rho}{\rho'}} \tag{2}$$

or conditions:

(1) Angles of climb equal,
(2) Angles of attack equal,

are satisfied all are satisfied. In the equations V is true air speed, v'_c the rate of climb, f is the power factor, HP is the power at sea-level conditions, and ρ is the air density. The primed and unprimed letters refer to two flight conditions of different pressure and temperature.

The use of a power factor is treated by H. Glauert in "A Discussion of the Law of Variation of Engine Power with Height," British Aeronautical Research Committee R. & M. No. 1099, issued in March, 1927. The conclusion given in the summary is: "The relative importance of pressure and density in determining the power of an engine appears to vary with height, and different methods of experiment lead to slightly discordant results. The simple pressure law is undoubtedly better than the simple density law, and for greater refinement Mr. Capon's suggestion should give a very close approximation to the truth."

It will be noted that up to this time no great effort has been made to find how the performance of an airplane actually varies with nonstandard conditions. In the pages following a logical method of reduction will be developed from the calculated effect of nonstandard conditions on performance.

OUTLINE OF METHODS AND BASIC CALCULATIONS

Preliminary studies indicated that the most direct method of attacking the problem of performance reduction was to calculate the effects of air pressure and temperature on the performance of a given airplane. This requires the adoption of (1) a relation between brake horsepower, air pressure, and air temperature, (2) a standard atmosphere, and (3) a specified variation of thrust horsepower with altitude in standard atmosphere.

In the present study the variation of B. HP. with P and T will be assumed to be (reference 1)—

$$\frac{\text{B.HP}}{\text{B.HP}_0}=\left(\frac{P}{P_0}\right)^{1.15}\left(\frac{T}{T_0}\right)^{-0.5} \tag{3}$$

The N. A. C. A. standard atmosphere has been officially adopted for general use in this country. Below 35,000 feet it is identical with the S. T. Ae. standard atmosphere used in Europe. The variation of thrust horsepower with altitude, given in Table I, has been obtained from analysis of flight test data by a simple method based on propeller power coefficients (reference 2). The data in Table I allow for effects of average change in R. P. M., B. HP., and propeller efficiency at constant true air speed.

The performance of a given airplane in standard atmosphere is readily calculated from the data given in Table I. At any given altitude in standard atmosphere for which the power curves have been obtained the pressure may be held constant and the effects of temperature on performance found by calculating new curves of power available from equation (3) and new curves of power required from the usual density relation $T.\ HP_2=T.\ HP_1\sqrt{\frac{\rho_1}{\rho_2}}$. This is equivalent to holding the pressure altitude constant while varying the density altitude. The results so obtained may be compared with the performance in normal standard atmosphere in order to build up an accurate method of performance reduction.

The first part of this investigation is based on the full scale polar for the VE-7 airplane as determined by Professor Lesley (reference 3). The effective aspect ratio of this airplane was taken as 4.8 in calculating the induced and parasite coefficients shown in Figure 1, which is based on Figure 11 of Lesley's report. The full scale data indicates a maximum lift coefficient of $C_{L\,max}=1.21$. The VE-7 has the following characteristics:

Wing area	289 sq. ft.
Wing section	R. A. F.-15.
Span, upper and lower	34.11 ft.
Gap	4.67 ft.
Chord, upper and lower	4.62 ft.

With a wing loading of 7.75 lb./sq. ft. the stalling speed at sea level is 50 mi./hr. This gives a gross weight of 2,230 pounds, which will be used throughout the remainder of the study.

Table II contains the air speeds and thrust horsepowers required at various altitudes, based on the data in Figure 1. Table III contains the calculations for thrust horsepower available at various altitudes, using the power factors from Table I. These data are plotted in Figure 2, from which the performance at altitude is obtained and summarized in Table IV. Figure 3 is a plot of rate of climb against altitude in normal standard atmosphere.

EFFECT OF TEMPERATURE ON PERFORMANCE AT 10,000 FEET PRESSURE ALTITUDE

The effect of temperature on performance was first investigated at 10,000 feet pressure altitude by assuming temperatures 83.3, 87.0, 90.9, 95.2, 1.05, 1.10, 1.15, and 1.20 per cent normal and calculating air speeds, powers required, and powers available. The air speeds

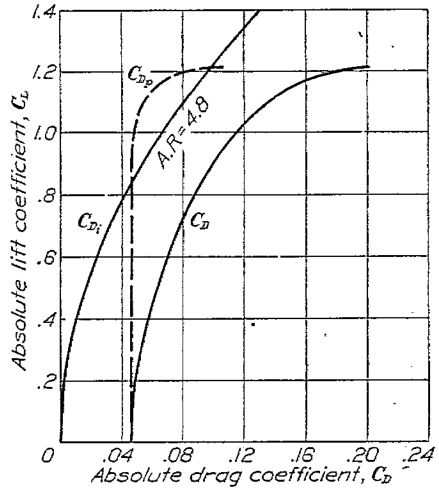

FIG. 1.—Full-scale polars of VE-7 airplane. Taken from Figure 11, N. A. C. A. Technical Report No. 220

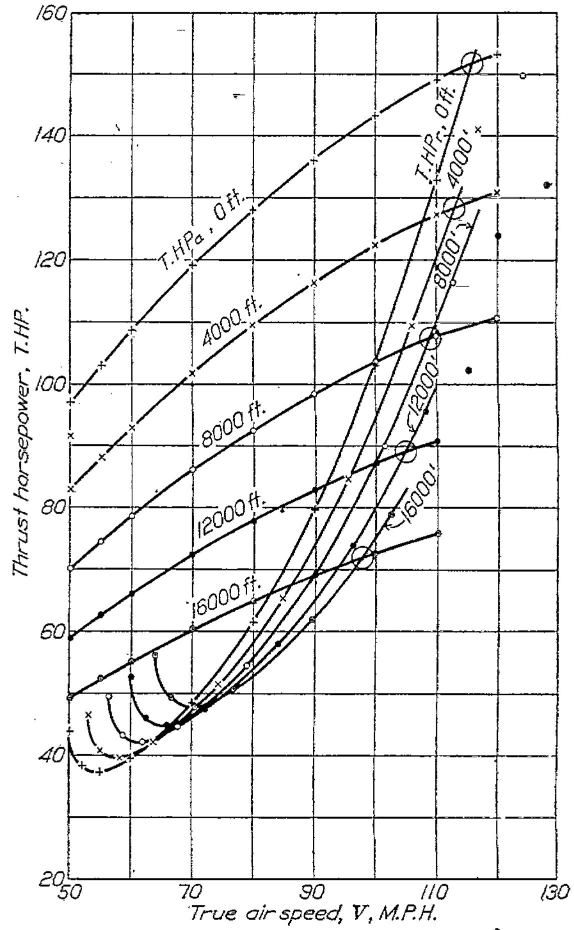

FIG. 2.—Power curves for gross weight of 2,230 pounds. VE-7 airplane. Normal standard atmosphere

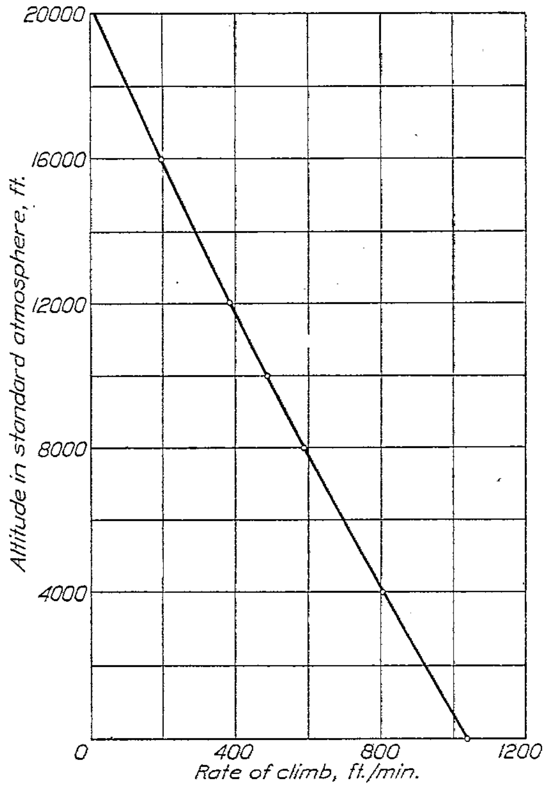

FIG. 3.—Rate of climb in standard atmosphere. Gross weight equals 2,230 pounds. VE-7 airplane

and powers required vary inversely as the square root of the density, while the powers available (at constant pressure) vary directly as the square root of the density. The particular low temperature ratios used are reciprocals of the corresponding high temperature ratios and were selected in order to simplify calculations while giving the same percentage change in density. The air speeds and powers required so calculated are given in Table V, and the powers available are given in Table VI. These data are plotted on two figures, 4 and 5, in order to avoid crowding the curves. Table VII contains a summary of the results obtained.

CLIMB

The altitude in normal standard atmosphere corresponding to the actual rate of climb for each temperature condition was read from Figure 3, and numerous methods of comparing these with the pressure and density altitudes were studied. The only method giving satisfactory results is that indicated in Table VII. The altitude in normal standard atmosphere at which the rate of climb is identical with that for a nonstandard temperature condition is given by

$$h = h_p - K\ (h_p - h_d) \tag{4}$$

Where h_p is the altitude in normal standard atmosphere having the observed pressure, h_d is the altitude in normal standard atmosphere having the observed density, and K is constant found in Table VII to be about 0.36. h_p and h_d are usually designated as pressure altitude and density altitude, respectively. The constant K does not show any dependence on the extent to which the temperatures depart from normal. In subsequent calculations, therefore, only the extreme values $\frac{T}{T_s}=0.833$ and $\frac{T}{T_s}=1.20$ need be considered.

MAXIMUM SPEEDS

As shown by Table VII, maximum speeds are not greatly affected by temperature changes. The small change indicated does not appear to bear any definite relation to the pressure altitude and density altitude considered together, as will be shown later. It is obvious, however, that the error involved in plotting true maximum air speeds against pressure altitude is less than the experimental error. It is also obvious that plotting maximum speeds against density altitude may lead to serious errors.

CLIMBING SPEEDS

Best climb appears to be at a substantially constant angle of attack or lift coefficient as shown by the slight variation in the ratio V_c/V_s, which shows a tendency to decrease slowly with increasing temperature. The maximum change in V_c/V_s likely to be encountered in practice is too small to be considered as a correction. A constant lift coefficient or angle of attack means a constant indicated air speed. Climbing speeds should therefore be specified in terms of indicated air speeds and aneroid pressures if this relation is found to hold for all altitudes.

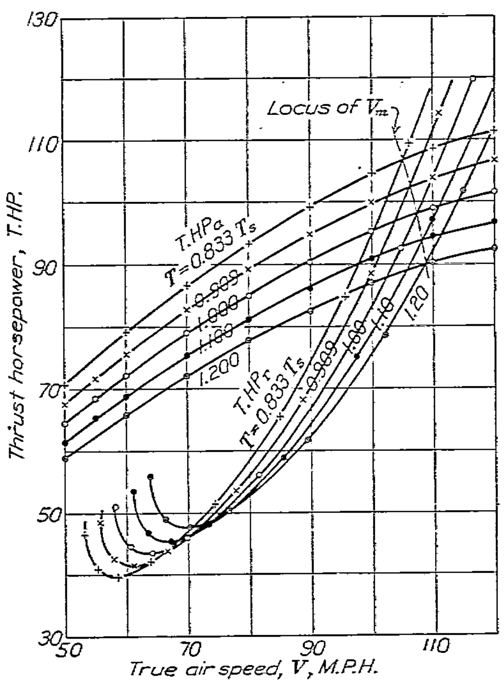

FIG. 4.—Power curves at 10,000-foot pressure altitude. Gross weight equals 2,230 pounds. See Figure 5, VE-7 airplane

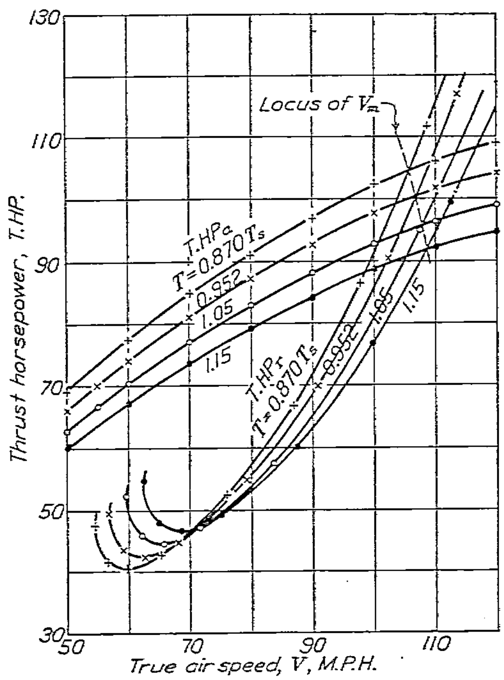

FIG. 5.—Power curves at 10,000-foot pressure altitude. Gross weight equals 2,230 pounds. See Figure 4, VE-7 airplane

EFFECT OF TEMPERATURE ON PERFORMANCE AT VARIOUS PRESSURE ALTITUDES

Following the method illustrated in Tables V and VI, powers required and powers available were calculated at pressure altitudes of 4,000, 8,000, 12,000, and 16,000 feet for temperatures 83.3, 90.9, 110, and 120 per cent normal. These calculations are too extensive to be included in the report, but the data are plotted in Figures 6, 7, 8, and 9. The indicated performances from these figures are given in Tables VIII and IX.

CLIMB

Values of the constant K in equation (4) show no marked dependence on altitude. A very slight tendency may be observed for K to increase with altitude, but the change is not appreciably greater than the accuracy of the calculations. For all practical purposes K may be considered independent of altitude.

MAXIMUM SPEED

The variation of maximum speed with temperature is slightly greater at low altitudes than at high altitudes, and has no simple dependence on the pressure altitude, density altitude, or any combination of the two. The extreme variation likely to be encountered at a given pressure altitude is about of the same order as the experimental error in determining the maximum speed. Maximum speeds should therefore be plotted against pressure altitude.

CLIMBING SPEEDS

At a given pressure altitude the ratio of climbing speed to stalling speed, V_c/V_s, decreases slightly with increase in temperature. This decrease is greater at low altitudes than at high altitudes, but the maximum change is comparatively small. Therefore at any given pressure altitude the angle of attack, the lift coefficient, and the indicated air speed for best climb are independent of the temperature.

VARIATION OF TEMPERATURE EFFECTS WITH CHANGES IN ASPECT RATIO AND PARASITE DRAG

Up to this point the investigation has been concerned with the performance of a single airplane, and some question naturally arises regarding the generality of the results so far obtained. It is proposed to give a convincing demonstration in the form of calculations covering a wide range in effective aspect ratio and parasite drag. For this purpose, a systematic series of 18 fictitious airplanes has been formed by combining 6 effective aspect ratios with 3 parasite drags. The weight, power, and wing area have been held constant at the VE–7 values for convenience, since the absolute values of these factors have no influence on the variation under investigation. The aspect ratios selected—3, 4, 5, 6, 8, and 10—completely cover the practical limits of present design. The parasite drag values adopted are based on the *normal* value of $C_{DP}=0.046$ shown on Figure 1 and include the *low* value $C_{DP}=0.031$ and the *high* value $C_{DP}=0.076$. The low value corresponds to about a 50 per cent in the structural parasite of the VE–7. The high value corresponds to about double the structural parasite of the VE–7.

Twenty-three groups of power required and power available curves, such as are shown on Figures 10, 11, and 12, were necessary to enable the accurate definition of the rate of climb curves in normal standard atmosphere as shown in Figures 13, 14, and 15. The results obtained in Tables VII, VIII, and IX indicate that calculations for two temperatures, one high and one low, at a single pressure altitude will be sufficient to determine the nature of the variation in K (equation 4). Accordingly, calculations were made for temperature ratios, $\frac{T}{T_s}=0.833$ and 1.20 at a pressure altitude of 10,000 feet for the normal and low parasite series, and at 8,000 feet for the high parasite series. The calculations are too extensive to be included in this report, but the power curve plots are given in Figures 16 to 21, inclusive. The data in Tables X, XI, and XII were obtained from Figures 16 to 21, using the same method previously employed. The results follow:

CLIMB

The values of K show a slight but very definite decrease with increase in aspect ratio and an indication of a slight increase with increase in parasite. The values of K for $T=1.20\ T_s$ are uniformly higher than for $T=0.833\ T_s$ by about 3 per cent. These relations are clearly brought out by the tabulation of K in Table XIII ano the plotting in Figure 22. The data do not enable any definite conclusion to be drawn in regard to the difference between the high and low temperature values of K, but there is no reason to suspect that it does not exist. The values at aspect ratio 5 check closely with those obtained for 4.8 in the first series, Tables VII, VIII, and IX. Since the maximum deviation from normal standard temperatures is probably less than 10 per cent, the value of K may be taken as 0.36 without appreciable error for any average airplane. For very high or very low aspect ratios the proper value from Figure 22 should be used.

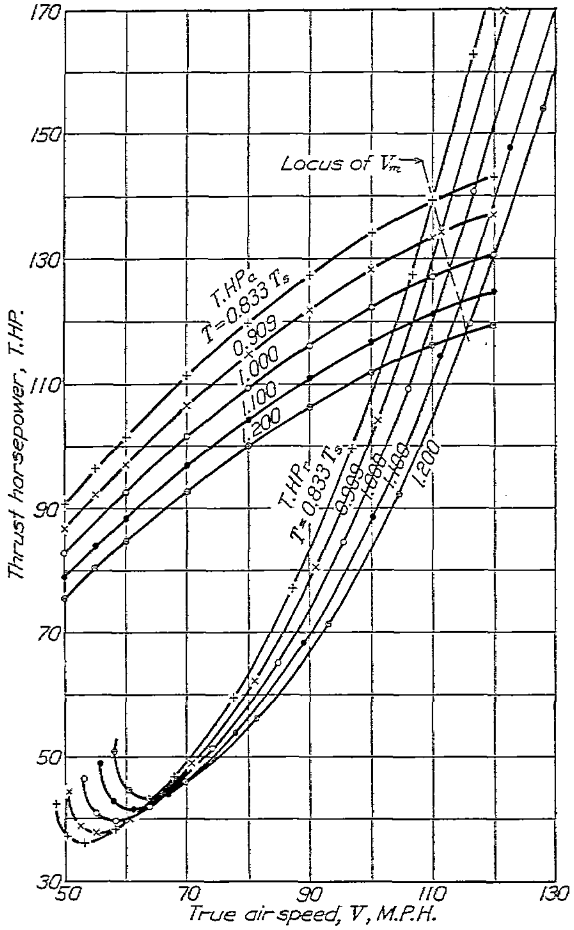

FIG. 6.—Power curves at 4,000-foot pressure altitude. Gross weight equals 2,230 pounds. VE-7 airplane

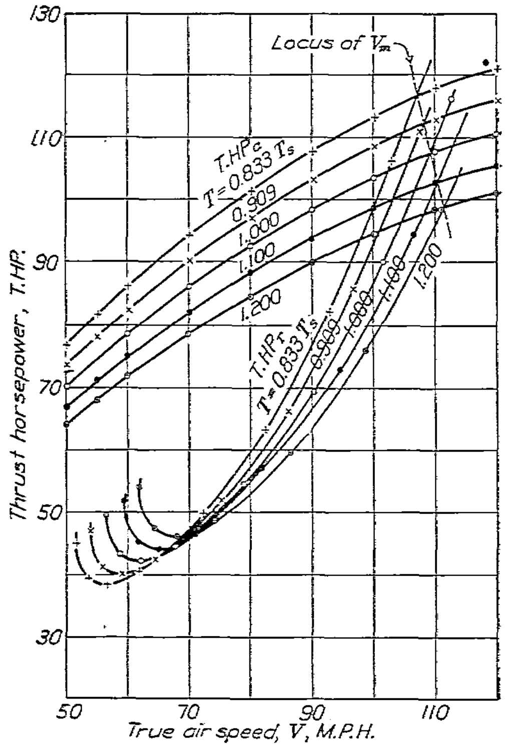

FIG. 7.—Power curves at 8,000-foot pressure altitude. Gross weight equals 2,230 pounds. VE-7 airplane

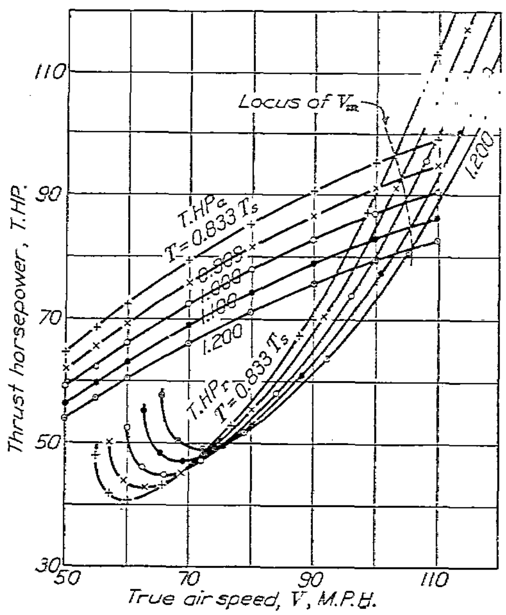

FIG. 8.—Power curves at 12,000-foot pressure altitude. Gross weight equals 2,230 pounds. VE-7 airplane

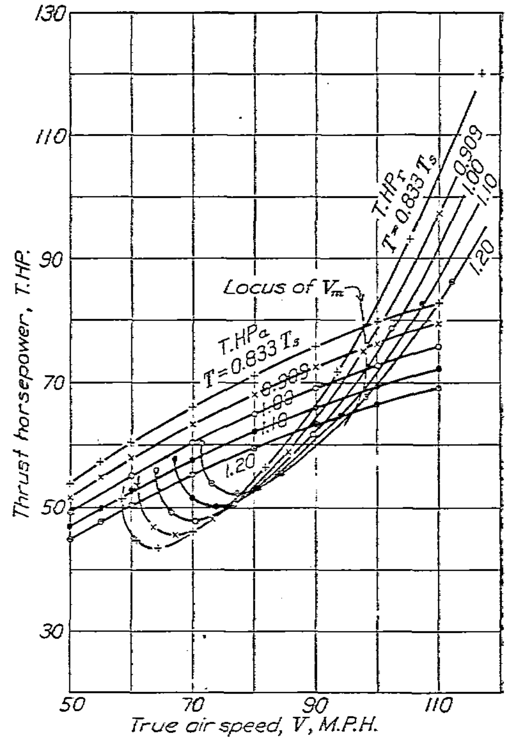

FIG. 9.—Power curves at 16,000-foot pressure altitude. Gross weight equals 2,230 pounds. VE-7 airplane

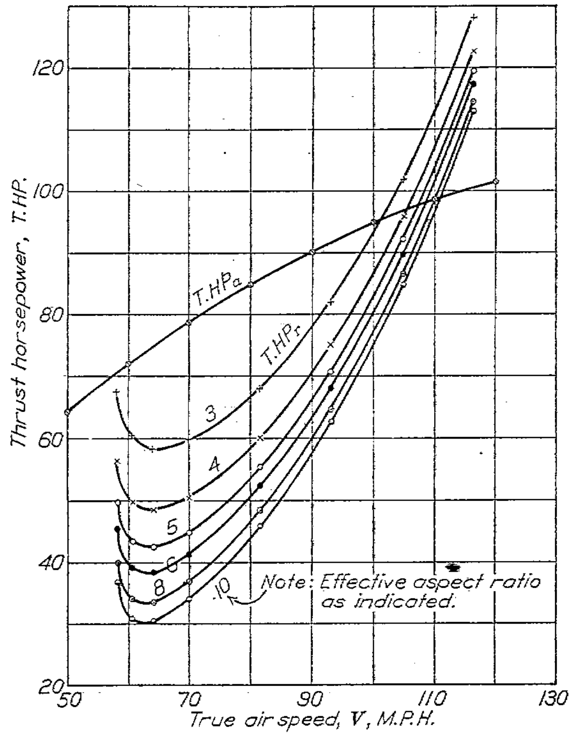

FIG. 10.—Power curves at 10,000 feet in normal standard atmosphere. *Normal parasite*

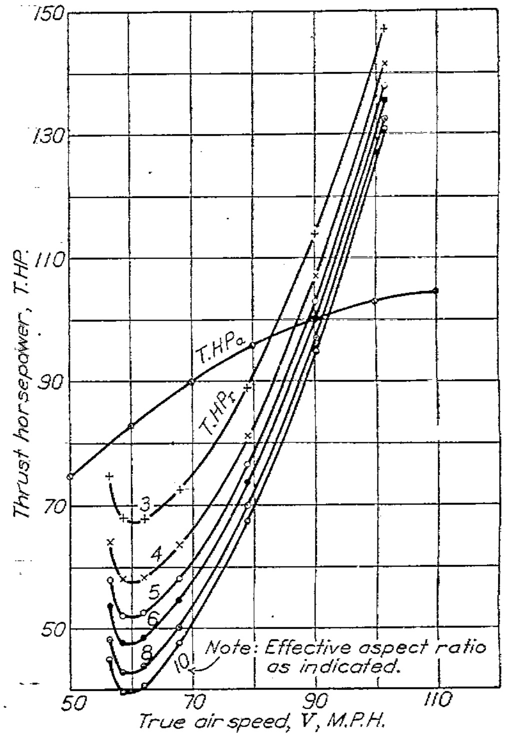

FIG. 12.—Power curves at 8,000 feet in normal standard atmosphere. *High parasite*

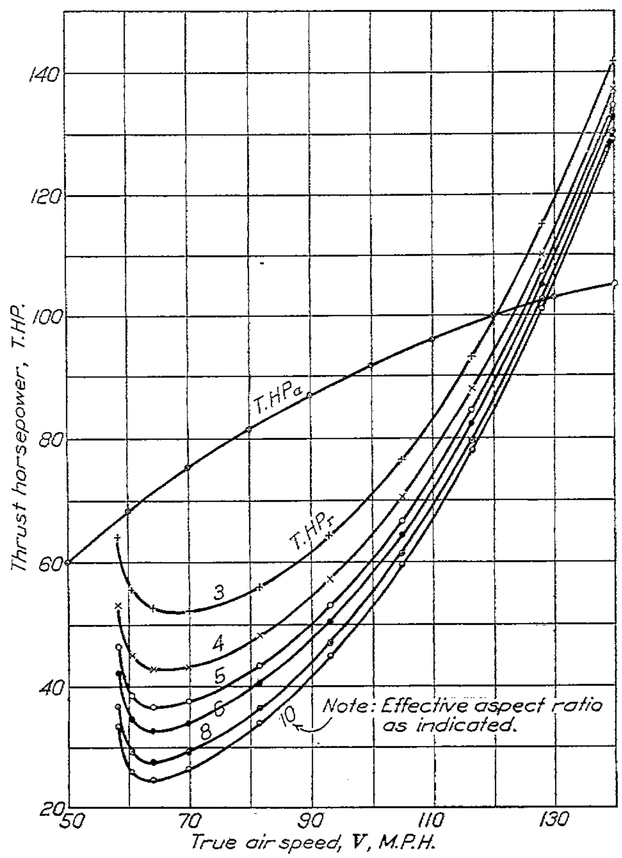

FIG. 11.—Power curves at 10,000 feet in normal standard atmosphere. *Low parasite*

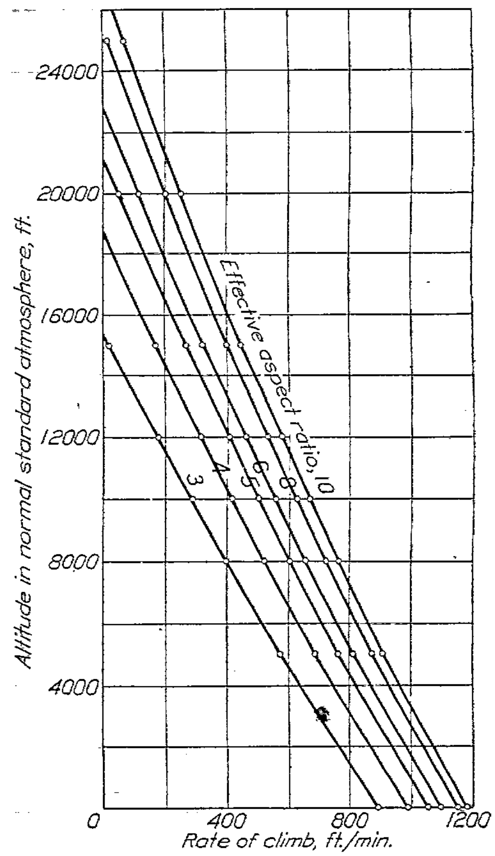

FIG. 13.—Rate of climb curves in normal standard atmosphere. *Normal parasite*

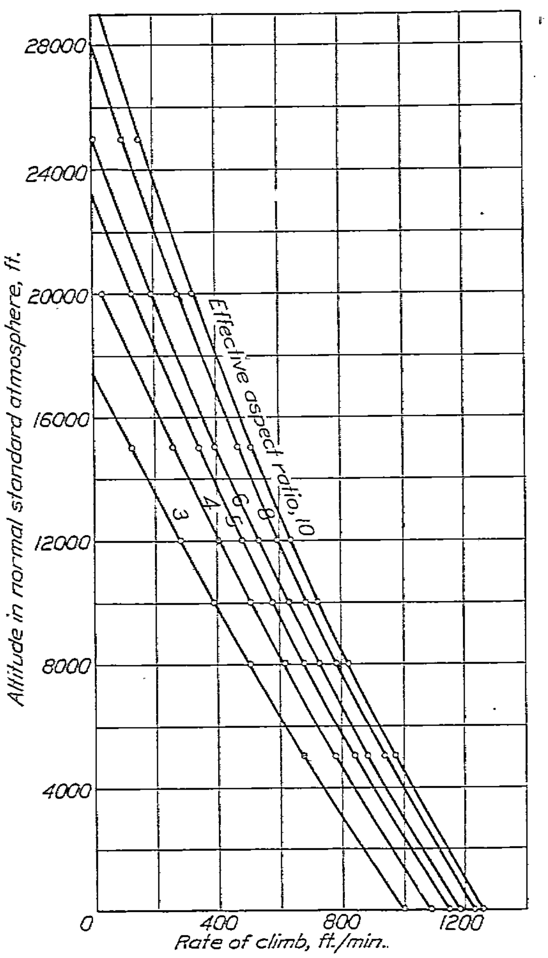

FIG. 14.—Rate of climb curves in normal standard atmosphere. *Low parasite*

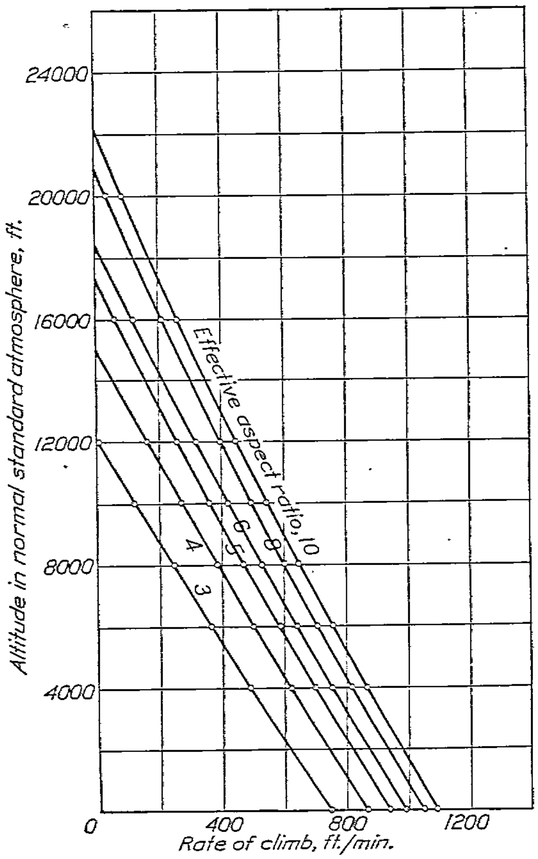

FIG. 15.—Rate of climb curves in normal standard atmosphere. *High parasite*

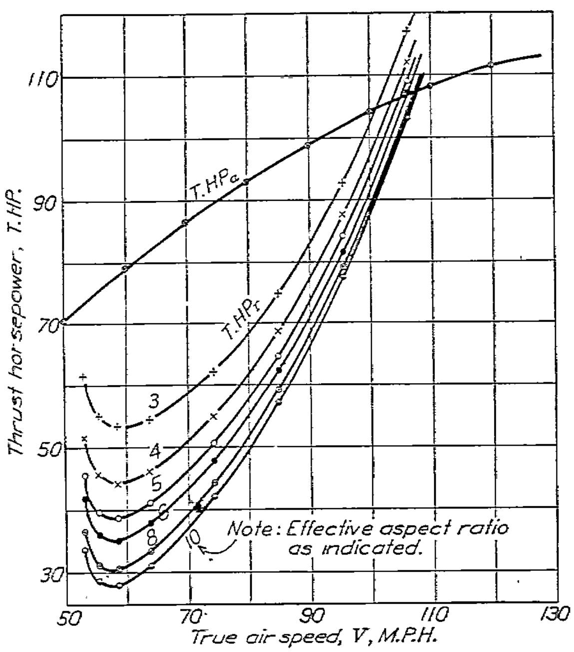

FIG. 16.—Power curves at 10,000-foot pressure altitude with temperature 83.3 per cent normal. *Normal parasite.* $T=0.833\ T_s$

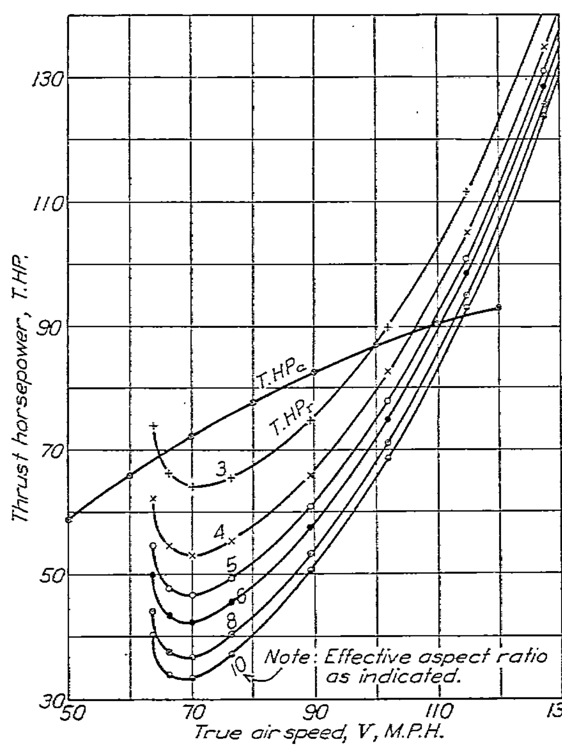

FIG. 17.—Power curves at 10,000-foot pressure altitude with temperatures 120 per cent normal. *Normal parasite.* $T=1.20\ T_s$

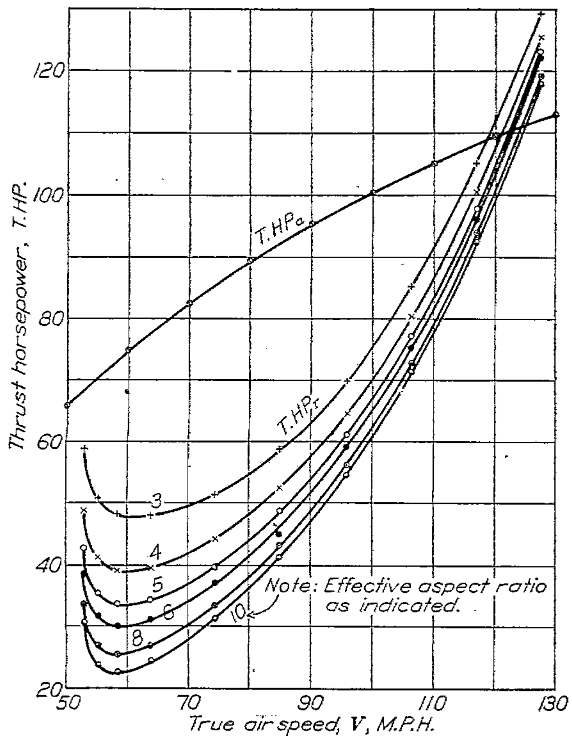

FIG. 18.—Power curves at 10,000-foot pressure altitude with temperature 83.3 per cent normal. *Low parasite.* $T=0.833\ T_s$

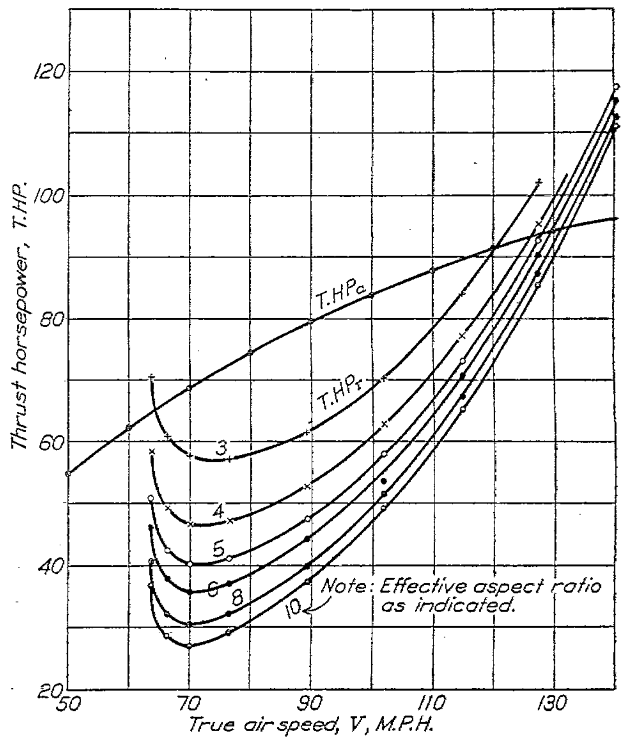

FIG. 19.—Power curves at 10,000-foot pressure altitude with temperature 120 per cent normal. *Low parasite.* $T=1.20\ T_s$

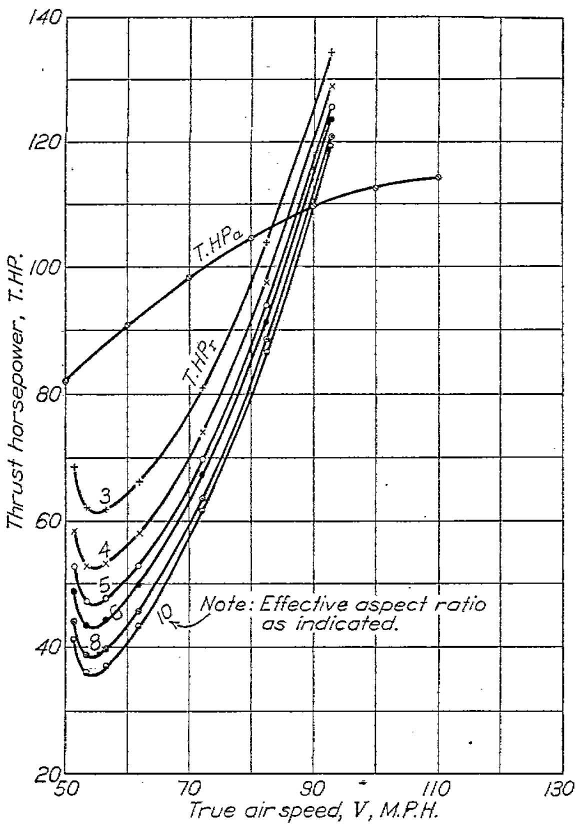

FIG. 20.—Power curves at 8,000-foot pressure altitude with temperature 83.3 per cent normal. *High parasite.* $T=0.833\ T_s$

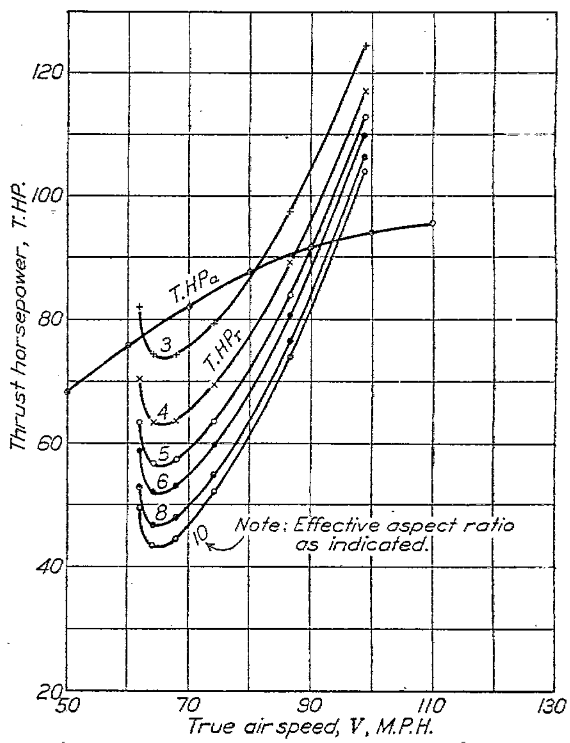

FIG. 21.—Power curves at 8,000-foot pressure altitude with temperature 120 per cent normal. *High parasite.* $T=1.20\ T_s$

MAXIMUM SPEED

The effect of temperature on maximum speed, at a given pressure altitude, increases with increase in aspect ratio and decreases with increase in parasite drag. The absolute change is always small and in practice the actual maximum speed should be plotted against pressure altitude. When great accuracy is required it would be reasonable to restrict speed runs to days on which the temperatures deviate less than 5 per cent from normal. There appears to be no practical method for applying the small correction to maximum speed.

CLIMBING SPEED

The effect of temperature on climbing speed at a given pressure altitude decreases with increase in parasite, but the change in indicated speed is so small that it can always be neglected.

DISCUSSION

It has been shown that the climbing performance of an airplane under nonstandard atmospheric conditions is comparable to the performance under standard conditions at an altitude h that is between the pressure altitude h_p and the density altitude h_d in accordance with the relation

$$h = h_p - K\ (h_p - h_d). \tag{4}$$

The value of K has been shown to be practically independent of altitude, temperature departure from normal, and parasite drag, but it is dependent to a noticeable degree on effective aspect ratio.

Two methods of reducing observed climb data to standard conditions are available. The proper value of K is selected from Table XIII or Figure 22, and the value of the "plotting altitude" or "equivalent altitude" h is calculated for set of pressure and temperature readings. The calculated rates of climb may then be plotted against h, or a curve of h against time may be drawn up and the rates of climb read from the slope of the curve. Both methods have given practically identical results in all cases investigated by the author. The first method appears to be the more accurate, while the second method requires less calculation in the reduction.

It has been shown that maximum speeds should be plotted against pressure altitudes. This entails a slight error, depending in magnitude upon the departure of the temperatures from normal. If great accuracy is desired, maximum speed runs should be restricted to days on which the temperatures are approximately normal.

It has also been shown that for a given airplane the indicated air speed for best climb at a given pressure altitude is practically independent of the temperature. Climbing speeds should therefore be specified as the variation of indicated air speed with pressure.

It is of interest to compare the new methods with the old. Referring to Table VII and Figure 3, it will be seen that for temperatures 10 per cent greater than normal at 10,000 feet pressure altitude the actual rate of climb is 60 ft./min. less than that at the pressure altitude and 95 ft./min. greater than that at the density altitude, while the maximum speed is 1.0 mi /hr. greater than that at the pressure altitude and 4.0 mi./hr. greater than that at the density altitude. The angle of attack for best climb is exactly the same as that at the pressure altitude and slightly less than that at the density altitude. The foregoing comparison shows clearly that the pressure and the density methods of reduction must be discarded as incorrect and unsatisfactory.

An analysis of the British method of reduction defined by equation (2) and accompanying text leads to the conclusion that the assumptions necessary for its validity are not fulfilled. This is particularly true for the condition of "equal angles of climb," since, as shown by Tables VII, VIII, and IX, for example, temperatures greater than normal increase the air speed for best climb but reduce the rate of climb, and vice versa.

Attention is invited to the fact that the methods of reduction developed in this report are based on the N. A. C. A. (or S. T. Ae.) standard atmosphere, which has been officially adopted by all interested Government organizations in this country.

The purpose of performance reduction has been stated before and it will be repeated for emphasis. Observed performance depends greatly on atmospheric conditions, and if the performances of two airplanes are to be compared they must be reduced to a common basis that eliminates the variation in atmospheric conditions. While any arbitrary atmosphere could be used in a reduction, the final results can not be compared in two different standard atmospheres.

An exact method of reduction will result in the same curve of rate of climb against altitude in standard atmosphere for a given airplane, regardless of atmosphere conditions under which the tests are made, provided that the temperatures are not beyond the capacity of the engine-cooling system. This does not mean that two reduced curves of altitude against time will be identical, or even very close together in the usual plotting. However, the reduced curves of altitude against time should differ only in a relative displacement along the time scale. In general, the density method of reduction results in a displacement along the altitude scale, while the pressure method of reduction results in relative rotation indicated by divergence of the two curves. The method outlined in this report appears to eliminate the divergence noted in pressure plotting, but, unfortunately, the author is unable at this time to secure good flight test data obtained under very extreme temperature conditions. However, the tendencies can be illustrated by the flight test data given and reduced in Tables XIV and XV. The data for the two climbs are not strictly comparable owing to a slight difference in the propeller blade setting, which accounts for part of the difference between the two climbs after reduction, but the plotted climbs on Figures 23, 24, and 25 bring out the characteristics of the three methods of reduction. Figure 23 shows the shifting of the curves along the altitude axis when the density method is used; Figure 24 shows the divergence characteristic of the pressure method, and Figure 25 shows how the new method eliminates the divergence observed in Figure 24. The vertical shift remaining in Figure 25 is due largely to the change in propeller blade setting, but the reduction in the vertical shift in comparison with Figure 23 is quite pronounced.

Bureau of Aeronautics,
Navy Department,
Washington, D. C., *May 18, 1928.*

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TABLE I

VARIATION OF THRUST HORSEPOWER WITH ALTITUDE IN STANDARD ATMOSPHERE AT CONSTANT AIR SPEED

Altitude, *h* feet	$\frac{T.\ HP}{T.\ HP_0}$	Altitude, *h* feet	$\frac{T.\ HP}{T.\ HP_0}$
0	1.000	16,000	0.508
2,000	.925	18,000	.465
4,000	.854	20,000	.425
5,000	.820	22,000	.387
6,000	.787	24,000	.352
8,000	.723	25,000	.336
10,000	.663	26,000	.320
12,000	.608	28,000	.291
14,000	.556	30,000	.261
15,000	.532		

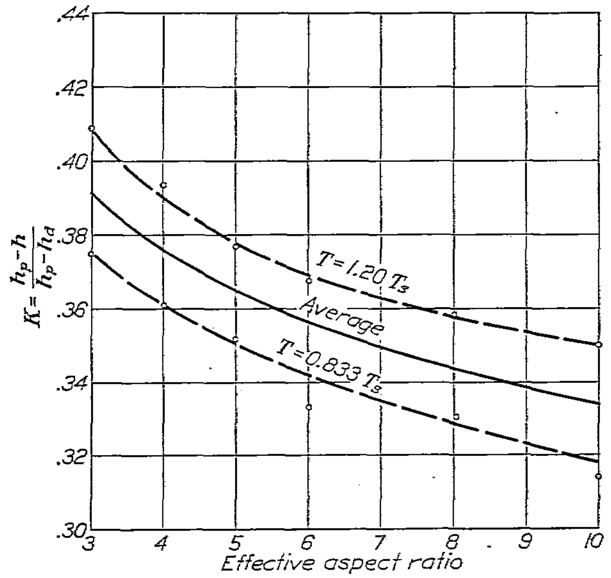

FIG. 22

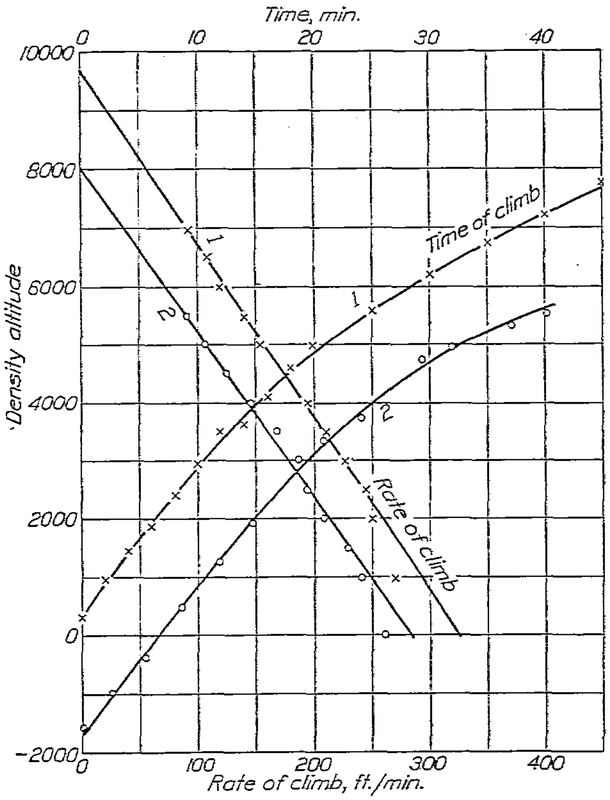

FIG. 23.—Reduction of climb by density method

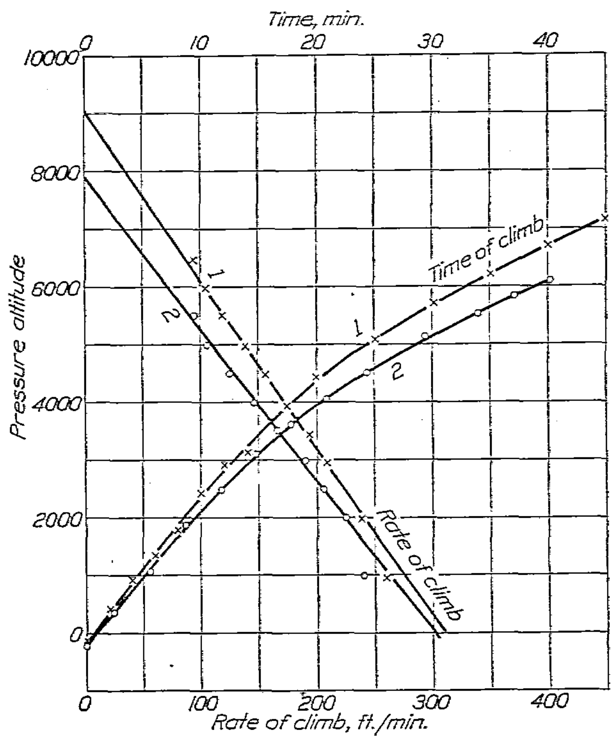

FIG. 24.—Reduction of climb by pressure method

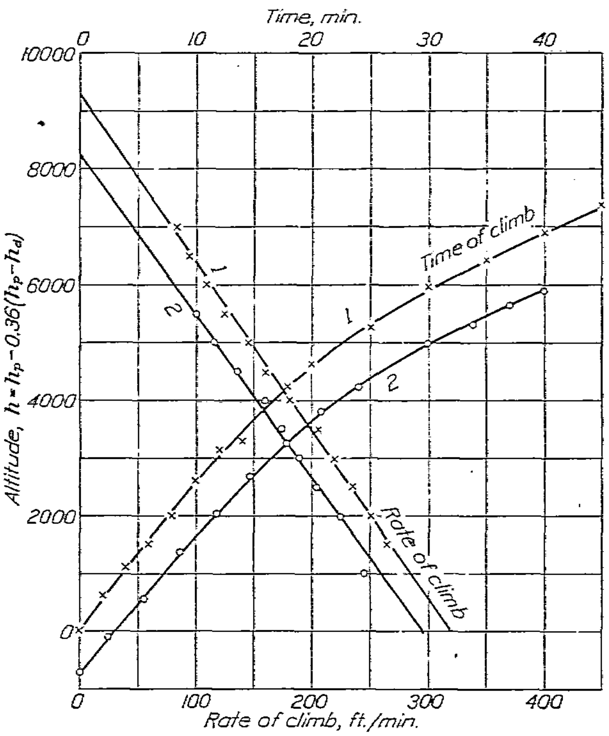

FIG. 25.—Reduction of climb by new method

TABLE II

VE-7 AIRPLANE—POWER REQUIRED FOR HORIZONTAL FLIGHT—GROSS WEIGHT, 2,230 POUNDS

Sea level		4,000 feet		8,000 feet		10,000 feet		12,000 feet		16,000 feet	
Air speed V mi./hr.	T. HP $_r$	V	T. HP $_r$	V	T. HP $_r$	V	T. HP $_r$	V	T. HP $_r$	V	T. HP $_r$
50	43.8	53.1	46.5	56.4	49.4	58.2	51.0	60.0	52.6	64.0	56.2
52	38.3	55.2	40.7	58.7	43.2	60.5	44.6	62.4	45.9	66.6	49.1
55	37.3	58.3	39.6	62.0	42.1	64.0	43.4	66.0	44.8	70.5	47.8
60	39.5	63.7	41.9	67.7	44.5	69.9	46.0	72.1	47.3	76.8	50.6
70	48.3	74.3	51.3	79.0	54.4	81.5	56.2	84.1	58.0	89.7	61.8
80	61.5	84.9	65.2	90.2	69.3	93.1	71.6	96.2	73.8	102.4	78.8
90	79.7	95.5	84.5	101.5	89.9	104.8	92.8	108.2	95.6	--------	--------
100	103.1	106.1	109.2	112.8	116.3	116.4	120.0	--------	--------	--------	--------
110	132.7	116.8	140.8	--------	--------	--------	--------	--------	--------	--------	--------
120	167.9	--------	--------	--------	--------	--------	--------	--------	--------	--------	--------

TABLE III

VE-7 AIRPLANE—THRUST HORSEPOWER AVAILABLE WITH WRIGHT E-4 ENGINE, 194 B. HP AT 1,800 R. P. M.

Air speed V mi./hr.	$\frac{V}{V_d}$	$\frac{\text{T. HP}}{\text{T. HP}_o}$	T. HP at sea level	T. HP at altitude				
				4,000 feet	8,000 feet	10,000 feet	12,000 feet	16,000 feet
50	0.434	0.640	97.0	82.8	70.1	64.5	59.0	49.2
60	.522	.716	108.6	92.7	78.5	72.2	66.0	55.2
70	.608	.785	119.0	101.6	86.0	79.1	72.4	60.4
80	.695	.845	128.0	109.3	92.5	85.2	77.8	65.0
90	.782	.898	136.0	116.2	98.3	90.4	82.7	69.1
100	.869	.945	143.2	122.3	103.5	95.3	87.1	72.7
110	.957	.985	149.0	127.2	107.7	99.1	90.6	75.7
115	1.000	1.000	151.4	130.7	110.6	101.7	--------	--------

TABLE IV

VE-7 AIRPLANE—CALCULATED PERFORMANCE IN STANDARD ATMOSPHERE—GROSS WEIGHT=2,230 POUNDS

Altitude feet	0	4,000	8,000	10,000	12,000	16,000
Stalling speed V_s mi./hr.	50.0	53.1	56.4	58.2	60.0	64.0
Max. speed V_m mi./hr.	116.0	113.2	109.2	107.3	105.0	98.6
Air speed for climb V_c mi./hr.	70.0	70.0	70.0	70.0	71.0	73.0
Excess T. HP for climb	70.5	54.7	40.0	32.8	26.0	13.0
Rate of climb	1,040	810	590	485	385	190

TABLE V

VE-7 AIRPLANE AT 10,000 FEET PRESSURE ALTITUDE—EFFECT OF TEMPERATURE ON V AND T. HP_r

V and T. HP_r for values of $\frac{T}{T_s}$ given							
0.833		0.870		0.909		0.952	
V	T. HP_r	V	T. HP_r	V	T. HP_r	V	T. HP_r
53.1	46.6	54.3	47.6	55.5	48.6	56.8	49.8
55.2	40.7	56.4	41.6	57.7	42.5	59.0	43.5
58.4	39.6	59.7	40.5	61.0	41.4	62.5	42.3
63.8	42.0	65.2	42.9	66.6	43.9	68.2	44.9
74.4	51.3	76.0	52.4	77.7	53.6	79.3	54.8
85.0	65.4	86.8	66.8	88.8	68.3	90.8	69.9
95.7	84.7	97.7	86.5	99.9	88.5	102.3	90.6
106.3	109.6	108.5	112.0	111.0	114.6	113.6	117.2
116.9	141.0	119.5	144.1	122.1	147.3	125.0	150.8

V and T. HP_r for values of $\frac{T}{T_s}$ given									
$\frac{T}{T_s}$=1.00		1.05		1.10		1.15		1.20	
V	T. HP_r	V	T. HP_r	V	T. HP_r	V	T. HP_r	V	T. HP_r
58.2	51.0	59.6	52.3	61.0	53.5	62.4	54.7	63.8	55.9
60.5	44.6	62.0	45.7	63.5	46.8	64.9	47.8	66.3	48.9
64.0	43.4	65.6	44.5	67.1	45.5	68.6	46.5	70.1	47.5
69.9	46.0	71.6	47.1	73.3	48.2	75.0	49.3	76.6	50.4
81.5	56.2	83.5	57.6	85.5	58.9	87.4	60.3	89.3	61.6
93.1	71.6	95.4	73.4	97.6	75.1	99.8	76.8	102.0	78.4
104.8	92.8	107.4	95.1	109.9	97.3	112.4	99.5	114.8	101.7
116.4	120.0	119.3	122.8	122.1	125.7	124.8	128.6	127.5	131.3
128.1	154.5								

TABLE VI

VE-7 AIRPLANE AT 10,000 FEET PRESSURE ALTITUDE—EFFECT OF TEMPERATURE ON T. HP AVAILABLE

T. HP_a									
$\frac{T}{T_s}$=	0.833	0.870	0.909	0.952	Normal 1.000	1.05	1.10	1.15	1.20
True air speed V mi./hr.:									
50	70.7	69.2	67.7	66.1	64.5	62.8	61.5	60.2	58.8
60	79.2	77.5	75.7	74.1	72.2	70.4	68.8	67.3	65.8
70	86.7	84.8	82.9	81.2	79.1	77.1	75.4	73.7	72.2
80	93.3	91.3	89.3	87.3	85.2	83.0	81.2	79.3	77.7
90	99.2	97.0	94.9	92.7	90.4	88.2	86.1	84.2	82.5
100	104.6	102.4	100.1	97.8	95.3	93.0	90.9	88.8	87.0
110	108.7	106.3	104.0	101.7	99.1	96.6	94.5	92.3	90.3
115	111.5	109.2	106.8	104.3	101.7	99.2	96.8	94.8	92.7

TABLE VII

VE-7 AIRPLANE AT 10,000 PRESSURE ALTITUDE—EFFECT OF AIR TEMPERATURE ON PERFORMANCE—GROSS WEIGHT=2,230 POUNDS

$\frac{T}{T_s}=$	0.833	0.870	0.909	0.952	1.00
Max. excess T. HP_a	40.2	38.6	36.7	25.1	33.2
Speed for climb, V_c	65.0	67.0	68.0	69.0	70.0
Max. speed, V_m mi./hr	105.2	105.4	106.2	106.8	107.3
Actual rate of climb, ft./min	595	571	543	520	492
Altitude in normal standard atmosphere for actual rate of climb, h ft	7,900	8,350	8,900	9,450	10,000
Density ratio, $\frac{\rho}{\rho_o}$	0.8861	0.8492	0.8122	0.7753	0.7384
Density altitude, h_d	4,080	5,480	6,930	8,430	10,000
$\Delta h = h_p - h$	2,100	1,650	1,100	550	0
$\Delta h_d = h_p - h_d$	5,920	4,520	3,070	1,570	0
$K = \Delta h / \Delta h_d$	0.355	0.365	0.358	0.351	-------
ΔV_m	−2.1	−1.9	−1.1	−.5	0
V_s	53.1	54.3	55.5	56.8	58.2
V_c/V_s	1.22	1.23	1.23	1.22	1.20

$\frac{T}{T_s}=$	1.05	1.10	1.15	1.20
Maximum excess T. HP_a	31.1	29.2	27.3	25.5
Speed for climb, V_c	72.0	73.0	74.0	75.0
Maximum speed, V_m mi./hr	107.8	108.3	108.5	108.8
Actual rate of climb, ft./min	460	432	404	377
Altitude in normal standard atmosphere for actual rate of climb, h ft	10,550	11,100	11,600	12,150
Density ratio, $\frac{\rho}{\rho_o}$	0.7032	0.6713	0.6421	0.6153
Density altitude, h_d	11,540	12,990	14,370	15,680
$\Delta h = h_p - h$	550	1,100	1,600	2,150
$\Delta h_d = h_p - h_d$	1,540	2,990	4,370	5,680
$K = \Delta h / \Delta h_d$	0.357	0.368	0.367	0.378
ΔV_m	+.5	+1.0	+1.2	+1.5
V_s	59.7	61.0	62.4	63.8
V_c/V_s	1.21	1.20	1.19	1.18

TABLE VIII

VE-7 AIRPLANE—EFFECT OF AIR TEMPERATURE ON PERFORMANCE AT 4,000 AND 8,000 FEET PRESSURE ALTITUDES

Pressure altitude	4,000 feet				
$\frac{T}{T_s}=$	0.833	0.909	1.00	1.10	1.20
Maximum excess T. HP_a	62.7	58.8	54.7	50.8	46.8
Speed for climb, V_c	65.0	67.5	70.0	71.0	72.0
Max. speed, V_m	110.2	111.6	113.2	114.6	115.6
Actual rate of climb, ft./min	928	870	810	752	693
Altitude in normal standard atmosphere for actual rate of climb, h ft	1,900	2,950	4,000	5,100	6,150
Density ratio, $\frac{\rho}{\rho_o}$	1.0657	0.9769	0.8881	0.8074	0.7400
Density altitude, h_d	−2,190	+790	4,000	7,130	9,930
$\Delta h=h_p-h$	2,100	1,050	0	1,100	2,150
$\Delta h_d=h_p-h_d$	6,190	3,220	0	3,130	5,930
$K=\Delta h/\Delta h_d$	0.349	0.326	----------	0.351	0.362
ΔV_m	−3.0	−1.6	0	1.2	2.4
V_s	48.5	50.6	53.1	55.7	58.2
V_c/V_s	1.34	1.34	1.32	1.28	1.24

Pressure altitude	8,000 feet				
$\frac{T}{T_s}=$	0.833	0.909	1.00	1.10	1.20
Maximum excess T. HP_a	47.5	43.7	40.0	36.0	32.5
Speed for climb, V_c	66.0	68.0	70.0	72.0	74.0
Maximum speed, V_m	107.2	108.2	109.2	110.5	111.3
Actual rate of climb, ft./min	703	647	592	533	481
Altitude in normal standard atmosphere for actual rate of climb, h ft	5,900	6,900	8,000	9,100	10,050
Density ratio, $\frac{\rho}{\rho_o}$	0.9431	0.8645	0.7859	0.7145	0.6549
Density altitude, h_d	1,980	4,880	8,000	11,030	13,760
$\Delta h=h_p-h$	2,100	1,100	0	1,100	2,050
$\Delta h_d=h_p-h_d$	6,020	3,120	0	3,030	5,760
$K=\Delta h/\Delta h_d$	0.349	0.352	----------	0.363	0.356
ΔV_m	−2.0	−1.0	0	1.3	2.1
V_s	51.5	53.8	56.4	59.2	61.9
V_c/V_s	1.28	1.26	1.24	1.22	1.20

TABLE IX

VE-7 AIRPLANE—EFFECT OF AIR TEMPERATURE ON PERFORMANCE AND 12,000 AND 16,000 FOOT PRESSURE ALTITUDES

Pressure altitude	12,000 feet				
$\frac{T}{T_s}=$	0.833	0.909	1.00	1.10	1.20
Maximum excess T. HP_a	33.3	30.0	26.0	22.6	19.3
Speed for climb, V_c	66.5	68.0	71.0	73.0	76.0
Maximum speed, V_m	103.3	104.2	104.7	105.6	105.7
Actual rate of climb, ft./min	493	443	385	334	285
Altitude in normal standard atmosphere for actual rate of climb, h ft	9,850	10,900	12,000	13,050	14,050
Density ratio, $\frac{\rho}{\rho_o}$	0.9317	0.7624	0.6931	0.6301	0.5776
Density altitude, h_d	6,160	8,970	12,000	14,950	17,590
$\Delta h=h_p-h$	2,150	1,100	0	1,050	2,050
$\Delta h_d=h_p-h_d$	5,840	3,030	0	2,950	5,590
$K=\Delta h/\Delta h_d$	0.368	0.363	----------	0.356	0.367
ΔV_m	−1.4	−.5	0	.9	1.0
V_s	54.8	57.2	60.0	62.9	65.7
V_c/V_s	1.19	1.19	1.18	1.16	1.16
Pressure altitude	16,000 feet				
$\frac{T}{T_s}=$	0.833	0.909	1.00	1.10	1.20
Maximum excess T. HP_a	20.0	17.0	13.0	9.8	6.7
Speed for climb, V_c	67.0	70.0	73.0	76.0	79.0
Maximum speed, V_m	97.8	98.2	98.6	97.5	95.8
Actual rate of climb, ft./min	295	250	197	147	99
Altitude in normal standard atmosphere for actual rate of climb, h ft	3,900	14,850	16,000	17,050	18,050
Density ratio, $\frac{\rho}{\rho_o}$	0.7306	0.6697	0.6088	0.5535	0.5073
Density altitude, h_d	10,340	13,060	16,000	18,860	21,430
$\Delta h=h_p-h$	2,100	1,150	0	1,100	2,050
$\Delta h_d=h_p-h_d$	5,560	2,940	0	2,860	5,430
$K=\Delta h/\Delta h_d$	0.371	0.391	----------	0.384	0.377
ΔV_m	−.8	−.4	0	−1.1	−2.8
V_s	58.4	61.0	64.0	67.1	70.1
V_c/V_s	1.15	1.15	1.14	1.13	1.13

TABLE X

EFFECT OF AIR TEMPERATURE ON PERFORMANCE OF AIRPLANES WITH NORMAL PARASITE DRAG AT 10,000 FEET PRESSURE ALTITUDE

Effective aspect ratio	3			4		
$\frac{T}{T_s}=$	0. 833	1. 00	1. 20	0. 833	1. 00	1. 20
Maximum excess T. HP	27. 7	19. 4	10. 6	35. 7	28. 3	20. 5
Speed for climb, V_c	69. 0	73. 0	79. 0	67. 0	72. 0	77. 0
Maximum speed, V_m	101. 2	102. 2	100. 0	103. 8	105. 7	106. 2
Actual rate of climb, ft./min	410	287	157	528	418	304
Altitude in normal standard atmosphere, for actual rate of climb, h ft	7, 800	10, 000	12, 300	7, 850	10, 000	12, 200
Density ratio	0. 8861	0. 7384	0. 6153	0. 8861	0. 7384	0. 6153
Density altitude, h_d	4, 080	10, 000	15, 680	4, 080	10, 000	15, 680
$\Delta h_d = h_p - h$	2, 200	0	−2, 300	2, 150	0	−2, 200
$\Delta h_d = h_p - h_d$	5, 920	0	−5, 680	5, 920	0	−5, 680
$K = \Delta h / \Delta h_d$	0. 372		0. 405	0. 363		0. 387
ΔV_m	−1. 0	0	−2. 2	−1. 9	0	+0. 5
V_s	53. 1	58. 2	63. 7	53. 1	58. 2	63. 7
V_c/V_s	1. 30	1. 25	1. 24	1. 26	1. 24	1. 21

Effective aspect ratio	5			6		
$\frac{T}{T_s}=$	0.833	1.00	1.20	0.833	1.00	1.20
Maximum excess T. HP	40. 9	34. 3	26. 8	44. 2	37. 8	30. 7
Speed for climb, V_c	65. 0	71. 0	75. 5	64. 5	70. 0	74. 0
Maximum speed, V_m	105. 2	107. 7	109. 1	106. 2	108. 9	110. 8
Actual rate of climb, ft./min	605	507	397	654	559	454
Altitude in normal standard atmosphere, for actual rate of climb, h ft	7, 950	10, 000	12, 200	8, 050	10, 000	12, 150
Density ratio	0. 8861	0. 7384	0. 6153	0. 8861	0. 7384	0. 6153
Density altitude, h_d	4, 080	10, 000	15, 680	4, 080	10, 000	15, 680
$\Delta h_d = h_p - h$	2, 050	0	−2, 200	1, 950	0	−2, 150
$\Delta h_d = h_p - h_d$	5, 920	0	−5, 680	5, 920	0	−5, 680
$K = \Delta h / \Delta h_d$	0. 346		0. 387	0. 329		0. 378
ΔV_m	−2. 3	0	+1. 4	−2. 7	0	+1. 9
V_s	53. 1	58. 2	63. 7	53. 1	58. 2	63. 7
V_c/V_s	1. 22	1. 22	1. 19	1. 21	1. 20	1. 16

Effective aspect ratio	8			10		
$\frac{T}{T_s}=$	0.833	1.00	1.20	0.833	1.00	1.20
Maximum excess T. HP	48. 4	42. 5	35. 8	50. 9	45. 2	39. 2
Speed for climb, V_c	63. 5	68. 0	73. 0	63. 0	67. 0	72. 7
Maximum speed, V_m	107. 6	110. 4	112. 8	108. 2	111. 2	114. 2
Actual rate of climb, ft./min	717	629	530	753	669	580
Altitude in normal standard atmosphere, for actual rate of climb, h ft	8, 100	10, 000	12, 050	8, 150	10, 000	12, 050
Density ratio	0. 8861	0. 7384	0. 6153	0. 8861	7, 384	6, 153
Density altitude, h_d	4, 080	10, 000	15, 680	4, 080	10, 000	15, 680
$\Delta h_d = h_p - h$	1, 900	0	−2, 050	1, 850	0	−2, 050
$\Delta h_d = h_p - h_d$	5, 920	0	−5, 680	5, 920	0	−5, 680
$K = \Delta h / \Delta h_d$	0. 321		0. 361	0. 313		0. 361
ΔV_m	−2. 8	0	+2. 4	−3. 0	0	+3. 0
V_s	53. 1	58. 2	63. 7	53. 1	58. 2	63. 7
V_c/V_s	1. 20	1. 17	1. 15	1. 19	1. 15	1. 14

TABLE XI

EFFECT OF AIR TEMPERATURE ON PERFORMANCE OF AIRPLANES WITH LOW PARASITE DRAG AT 10,000 FEET PRESSURE ALTITUDE

Effective aspect ratio	3			4		
$\frac{T}{T_s}=$	0.833	1.00	1.20	0.833	1.00	1.20
Maximum excess T. HP	34.5	26.2	17.7	41.5	34.3	26.7
Speed for climb, V_c	78.5	82.0	88.5	76.0	80.0	86.0
Maximum speed, V_m	118.7	120.2	120.6	121.0	123.8	126.4
Actual rate of climb, ft./min	511	387	262	614	507	395
Altitude in normal standard atmosphere, for actual rate of climb, h ft	7,850	10,000	12,350	7,950	10,000	12,200
Density ratio	0.8861	0.7384	0.6153	0.8861	0.7384	0.6153
Density altitude, h_d	4,080	10,000	15,680	4,080	10,000	15,680
$\Delta h=h_p-h$	2,150	0	−2,350	2,050	0	−2,200
$\Delta h_d=h_p-h_d$	5,920	0	−5,680	5,920	0	−5,680
$K=\Delta h/\Delta h_d$	0.363	------	0.414	0.346	------	0.387
ΔV_m	−1.5	0	+0.4	−2.8	0	+2.6
V_s	53.1	58.2	63.7	53.1	58.2	63.7
V_c/V_s	1.48	1.41	1.39	1.43	1.38	1.35

Effective aspect ratio	5			6		
$\frac{T}{T_s}=$	0.833	1.00	1.20	0.833	1.00	1.20
Maximum excess T. HP	46.0	39.0	32.3	49.0	42.8	36.0
Speed for climb, V_c	74.0	78.0	85.0	72.5	76.0	82.0
Maximum speed, V_m	122.5	125.4	128.1	123.1	126.5	129.8
Actual rate of climb, ft./min	681	577	478	725	633	533
Altitude in normal standard atmosphere, for actual rate of climb, h ft	7,950	10,000	12,100	8,100	10,000	12,050
Density ratio	0.8861	0.7384	0.6153	0.8861	0.7384	0.6153
Density altitude, h_d	4,080	10,000	15,680	4,080	10,000	15,680
$\Delta h=h_p-h$	2,050	0	−2,100	1,900	0	−2,050
$\Delta h_d=h_p-h_d$	5,920	0	−5,680	5,920	0	−5,680
$K=\Delta h/\Delta h_d$	0.346	------	0.370	0.321	------	0.361
ΔV_m	−2.9	0	+2.7	−3.4	0	+3.3
V_s	53.1	58.2	63.7	53.1	58.2	63.7
V_c/V_s	1.39	1.34	1.33	1.36	1.31	1.29

Effective aspect ratio	8			10		
$\frac{T}{T_s}=$	0.833	1.00	1.20	0.833	1.00	1.20
Maximum excess T. HP	52.5	46.5	40.3	54.7	49.0	43.4
Speed for climb, V_c	71.0	75.0	80.0	69.0	74.0	79.0
Maximum speed, V_m	124.4	128.0	131.3	125.0	128.6	132.5
Actual rate of climb, ft./min	777	688	596	810	725	642
Altitude in normal standard atmosphere, for actual rate of climb, h ft	8,100	10,000	12,000	8,200	10,000	11,900
Density ratio	0.8861	0.7384	0.6153	0.8861	0.7384	0.6153
Density altitude, h_d	4,080	10,000	15,680	4,080	10,000	15,680
$\Delta h=h_p-h$	1,900	0	−2,000	1,800	0	−1,900
$\Delta h_d=h_p-h_d$	5,920	0	−5,680	5,920	0	−5,680
$K=\Delta h/\Delta h_d$	0.321	------	0.352	0.304	------	0.335
ΔV_m	−3.6	0	+3.3	−3.6	0	+3.9
V_s	53.1	58.2	63.7	53.1	58.2	63.7
V_c/V_s	1.34	1.29	1.26	1.30	1.27	1.24

TABLE XII

EFFECT OF AIR TEMPERATURE ON PERFORMANCE OF AIRPLANES WITH HIGH PARASITE DRAG, 8,000 FEET PRESSURE ALTITUDE

Effective Aspect Ratio	3			4		
$\frac{T}{T_s}=$	0.833	1.00	1.20	0.833	1.00	1.20
Maximum excess T. HP	26.3	16.9	6.8	34.8	26.3	17.2
Speed for climb, V_c	59.0	64.0	69.0	58.2	64.0	67.7
Maximum speed, V_m	83.6	83.2	81.1	85.9	87.4	87.5
Actual rate of climb, ft./min	389	250	101	51.5	389	255
Altitude in normal standard atmosphere, for actual rate of climb, h ft	5,650	8,000	10,350	5,750	8,000	10,350
Density ratio	0.9431	0.7859	0.6549	0.9431	0.7859	0.6549
Density altitude, h_d	1,980	8,000	13,760	1,980	8,000	13,760
$\Delta h=h_p-h$	2,350	0	−2,350	2,250	0	−2,350
$\Delta h_d=h_p-h_d$	6,020	0	−5,760	6,020	0	−5,760
$K=\Delta h/\Delta h_d$	0.390	---	0.408	0.374	---	0.408
ΔV_m	+0.4	0	−2.1	−1.5	0	+0.5
V_s	51.5	56.4	61.8	51.5	56.4	61.8
V_c/V_s	1.15	1.13	1.12	1.13	1.13	1.10

Effective aspect ratio	5			6		
$\frac{T}{T_s}=$	0.833	1.00	1.20	0.833	1.00	1.20
Max. excess T. HP	40.2	31.8	23.9	43.7	35.8	28.2
Speed for climb, V_c	57.7	63.5	67.5	57.5	62.5	67.2
Max. speed, V_m	87.3	89.3	90.2	88.4	90.6	91.9
Actual rate of climb, ft./min	595	470	354	647	529	418
Altitude in normal standard atmosphere, for actual rate of climb, h ft	5,800	8,000	10,150	5,900	8,000	10,100
Density ratio	0.9431	0.7859	0.6549	0.9431	0.7859	0.6549
Density altitude, h_d	1,980	8,000	13,760	1,980	8,000	13,760
$\Delta h=h_p-h$	2,200	0	−2,150	2,100	0	−2,100
$\Delta h_d=h_p-h_d$	6,020	0	−5,760	6,020	0	−5,760
$K=\Delta h/\Delta h_d$	0.365	---	0.373	0.349	---	0.364
ΔV_m	−2.0	0	+0.9	−2.2	0	+1.3
V_s	51.5	56.4	61.8	51.5	56.4	61.8
V_c/V_s	1.12	1.13	1.09	1.12	1.11	1.09

Effective aspect ratio	8			10		
$\frac{T}{T_s}=$	0.833	1.00	1.20	0.833	1.00	1.20
Max. excess T. HP	48.3	40.8	33.3	50.8	43.7	36.7
Speed for climb, V_c	56.5	62.0	67.2	56.0	62.0	67.2
Max. speed, V_m	89.4	91.9	93.7	90.0	92.7	94.8
Actual rate of climb, ft./min	715	603	494	752	646	543
Altitude in normal standard atmosphere, for actual rate of climb, h ft	5,900	8,000	10,100	6,050	8,000	10,050
Density ratio	0.9431	0.7859	0.6549	0.9431	0.7859	0.6549
Density altitude, h_d	1,980	8,000	13,760	1,980	8,000	13,760
$\Delta h=h_p-h$	2,100	0	−2,100	1,950	0	−2,050
$\Delta h_d=\Delta h_p-h_d$	6,020	0	−5,760	6,020	0	−5,760
$K=\Delta h/\Delta h_d$	0.349	---	0.364	0.324	---	0.356
ΔV_m	−2.5	0	+1.8	−2.7	0	+2.1
V_s	51.5	56.4	61.8	51.5	56.4	61.8
V_c/V_s	1.10	1.10	1.09	1.09	1.10	1.09

TABLE XIII

Altitude correction factor $K=\frac{h_p-h}{h_p-h_d}$

SUMMARY OF RESULTS FROM TABLES X, XI, AND XII

$\left.\frac{\text{Temperature}}{\text{Normal Temperature}}\right\}=$	0.833			1.20		
Parasite drag	Normal	Low	High	Normal	Low	High
Effective aspect Ratio 3	0.372	0.363	0.390	0.405	0.414	0.408
4	.363	.346	.374	.387	.387	.408
5	.346	.346	.365	.387	.370	.373
6	.329	.321	.349	.378	.361	.364
8	.321	.321	.349	.361	.352	.364
10	.313	.305	.324	.361	.335	.355

TABLE XIV

EXAMPLE OF REDUCTION OF CLIMB DATA—CLIMB NO. 1

Time	Pressure	Temperature °C.	Altitudes (feet)				
			Pressure h_p	Density h_d	h_p-h_d, Δh	0.36 Δh	Plotting altitude, h
0	765	20	−180	+340	−520	−190	+10
2	749	19	400	970	−570	−210	610
4	735	18	920	1,460	−540	−190	1,110
6	724	17	1,340	1,880	−540	−190	1,530
8	712	16.5	1,790	2,420	−640	−230	2,020
10	696	15	2,410	2,970	−560	−200	2,610
12	683	14	2,920	3,520	−600	−220	3,140
14	678	13	3,120	3,630	−510	−180	3,300
16	668	13	3,530	4,110	−580	−210	3,740
18	656	12	4,010	4,610	−600	−220	4,230
20	646	11	4,430	4,980	−550	−200	4,630
25	630	9	5,100	5,570	−470	−170	5,270
30	616	8	5,700	6,190	−490	−186	5,980
35	604	7	6,220	6,740	−520	−190	6,410
40	593	6	6,700	7,230	−530	−190	6,890
45	583	6	7,150	7,790	−640	−230	7,380

TABLE XV

EXAMPLE OF REDUCTION OF CLIMB DATA—CLIMB NO. 2

Time	Pressure	Temperature °C.	Altitudes (feet)				
			Pressure altitude, h_p	Density altitude, h_d	$h_p - h_d$, Δh	$0.36\ \Delta h$	Plotting altitude, h
0	766	4	−220	−1, 580	+1, 360	490	−710
2. 48	750	3	+370	−980	+1, 350	490	−120
5. 58	731	1	1, 070	−370	+1, 440	520	+550
8. 68	710	0	1, 870	+480	+1, 390	500	1, 370
11. 80	694	0	2, 490	1, 250	+1, 240	450	2, 040
14. 72	678	−1	3, 120	1, 920	+1, 200	430	2, 690
17. 82	666	+1	3, 610	2, 820	+990	360	3, 250
20. 92	655	+1	4, 050	3, 330	+720	260	3, 790
24. 02	644	0	4, 510	3, 730	+780	280	4, 230
29. 28	629	0	5, 140	4, 730	+410	150	4, 990
33. 93	620	−1	5, 520	4, 950	+570	210	5, 310
37. 03	613	−1	5, 830	5, 330	+500	180	5, 650
40. 14	607	−2	6, 090	5, 530	+560	200	5, 890